GUIDANCE AND NAVIGATION AT MARCOPOLO-R ASTEROID 2008EV5

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Abstract: *This paper aims at presenting the results obtained from the analysis and simulation of the Guidance and Navigation performances for the proximity phase at asteroid 2008 EV5 in the MarcoPolo-R mission study. The activities carried out in this task were key in supporting the design of the mission GNC and operations architecture. Simulations were executed to assess the performances achievable in the estimation of the spacecraft asteroid relative state, of the asteroid ephemerides, and of the asteroid gravitational parameter. Monte Carlo simulations were run to assess the guidance performances for polar global observation orbits, inertial hovering, and radial descent phase.*

Keywords: *Minor bodies, guidance, navigation, MarcoPolo-R*

1. Introduction

MarcoPolo-R was a candidate ESA asteroid sample return mission competing in the Cosmic Vision programme for the M3 launch opportunity. The aim of the mission was to reach a primitive Near-Earth Asteroid (NEA) and retrieve material from it, to be delivered back to Earth for study. After completing the interplanetary transfer and the asteroid proximity operations, the spacecraft would follow a return trajectory to Earth, culminating in atmospheric re-entry, and finally landing in the Woomera Test Range.

In the frame of the study Elecnor Deimos, member of the consortium led by Thales Alenia Space–Italy under ESA contract, was responsible for interplanetary transfer design and GNC, proximity phase trajectory, Earth entry descent and landing analysis and guidance and operations design; this paper will focus on the main results obtained regarding the initial assessment of the asteroid proximity trajectory GNC performances.

2. Reference orbital condition

Different flight conditions can be employed during the asteroid proximity phase to achieve the goals of the mission. For the navigation analysis presented in the following, the reference flight condition will be a Polar Controlled Orbit. For this type of motion the orbit is almost circular and its plane is polar with respect to the plane of motion of the asteroid around the Sun; normally deterministic control is required to maintain size, inclination and solar time of the ascending node in the dynamic environment provided by the asteroid gravitational acceleration and the solar third body and radiation pressure perturbations.

Unless specified otherwise, a reference altitudes of 5 km with respect to the asteroid will be considered for the navigation analysis; the orbital period for such altitude would be 10.7 days approximately.

The specificities of the design of controlled polar orbits is addressed in [2] and [3], and will not be treated here, not being the details of the orbital motion at minor bodies the focus of this paper. The design of an actual operational scenario was not carried out during the study. As a consequence, the reference orbital condition taken here is meant to allow assessing the indicative guidance and navigation performances at asteroid proximity, and not to provide actual expected performances for a specific scenario.

The target body for the mission at the time of the execution of the analyses presented here was 2008 EV5. Making reference to the observations and analyses of Busch et al. (see [1]), the main characteristics of the asteroid assumed for the following analyses is summarised in Table 1.

Table 1. Asteroid 2008 EV5 assumed properties

2008 EV5	
Shape	Spherical
Radius	225 m
Mass	10^{11} kg
Gravitational parameter	6.674×10^{-9} km^3/kg/s^2

3. Navigation at asteroid proximity

The navigation performances achievable at asteroid proximity were analysed. The main assumptions employed for the navigation simulations are summarised in Table 2. Continuous tracking from three ground stations was assumed in order to assess what are the best estimation performances achievable as function of the orbital configuration and available observables. In real operations the tracking schedule will be accommodated to fit also other operational needs (observation, commanding,...), whose definition is not the purpose of this paper. The results provided will allow nevertheless an assessment of the achievable OD performances during the proximity flight phase. Notice that the altimetry error budget is conservative, and is meant to include the uncertainty in the knowledge of the asteroid shape.

The estimation process is assessed making use of covariance analysis simulations, carried out with a Square Root Information Filter described by Bierman in [4] and in the implementation described in [5].

The Controlled Polar Orbit with 5 km altitude previously described was used. Figure 1 shows the Earth-Asteroid-S/C geometry for the simulated condition.

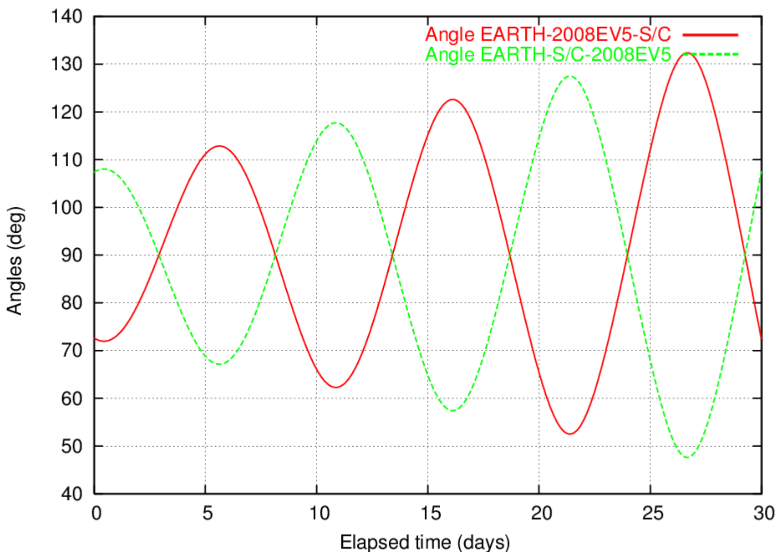

Figure 1. Earth-S/C geometry for navigation simulation

Table 2. Navigation assumptions summary

Radiometric Measurements		
	Noise (1-sigma)	Bias (1-sigma)
Range	4 m	20 m
Doppler	0.075 mm/s	-
Ground station position (error per component)	-	50 cm
GS Considered	Cebreros, New Norcia, Malargüe	
Acquisition frequency	Range every 2 hours, Doppler every 20 minutes	
Optical Measurements		
Field Of View	50 deg, 1024 px	
Measurement errors	1 px noise (1-sigma), 8 μrad bias (1-sigma)	
Acquisition frequency	One every 20 minutes	
Physical environment uncertainties		
Asteroid initial ephemeris error	1000 km, 1-sigma, in each direction	
Asteroid grav. parameter error	$5x10^{-9}$ km^3/s^2 (50% of nominal value, according to [1])	
Non-Gravitational Accelerations	$5x10^{-11}$ km/s^2 in each direction, 1 day autocorrelation time	
Altimetry measurements		
Measurement error	1% (1-sigma)	
Acquisition frequency	As optical data	
Initial Conditions		
Initial S/C knowledge error	1 km and 10 mm/s, 1-sigma, in each direction	

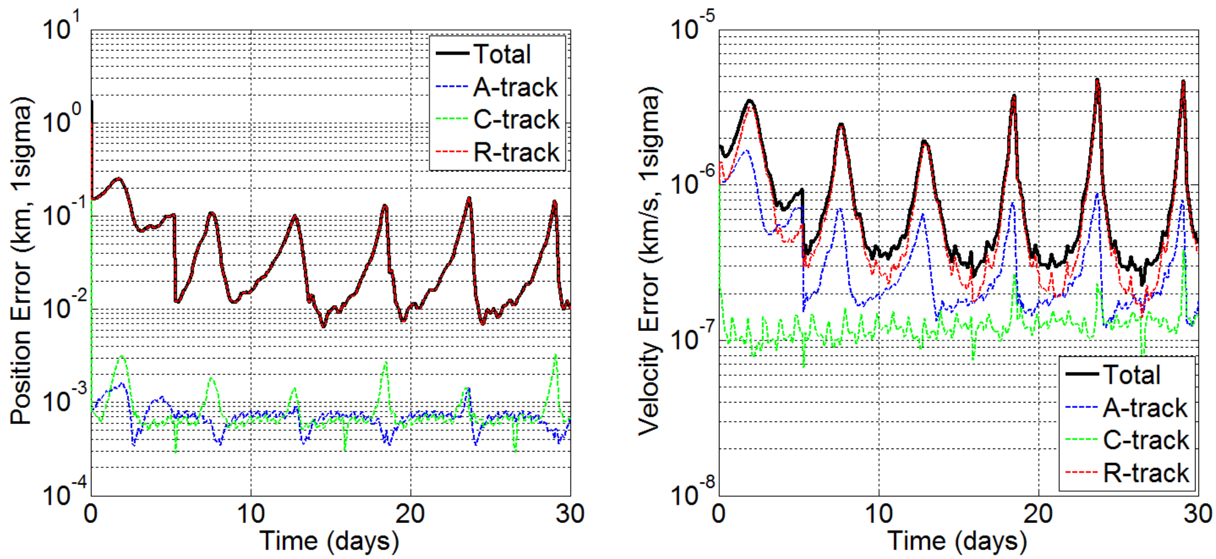

Figure 2. Asteroid-S/C state estimation with radiometric tracking and LoS measurements

Ground-based radiometric alone would not be sufficient to estimate the asteroid-S/C state with a precision sufficient to successfully carry out the proximity phase thus, as expectable, optical measurements providing direct information regarding the asteroid-S/C state are required to achieve levels of accuracy in state knowledge able to ensure feasibility of the proximity flight.

Figure 2 shows the OD performances obtained making use of radiometric data plus optical measurements to the asteroid obtained centre of brightness. It is assumed that optical measurements are Line of Sight (LoS) angles to the centre of the asteroid. The centre estimation is assumed to be possible with an error of 1px (1-sigma); a *considered* bias of 8 μrad is assumed for optical measurements, in order to represent uncertainties due, among others, to the pointing of the navigation camera and to the spacecraft attitude determination.

LoS measurements provide direct information on the position of the S/C in a plane perpendicular to the radial direction to the asteroid centre. As a consequence, in Figure 2 it is possible to see that the estimation of the along-track and cross-track position is in the order of 1 m (1-sigma).

The radial component, on the other hand, can be estimated only through radiometric data, and it is related to the signature left on such data by the gravitational acceleration of the asteroid. The estimation accuracy in range is in the order of 10 to 100 m (1-sigma). The velocity error is in the order of 0.1 mm/s for the cross-track component, directly estimated through the optical measurements. The along-track and radial velocity components are related to the distance of the S/C from the asteroid and to its gravitational acceleration, and as such are observable through radiometric measurements.

The peaks in range error and in radial and along-track velocity are due to the fact that, in the case considered, at those times the S/C-Asteroid-Earth angle is 90 deg, and in such conditions the observability provided by radiometric tracking is degraded.

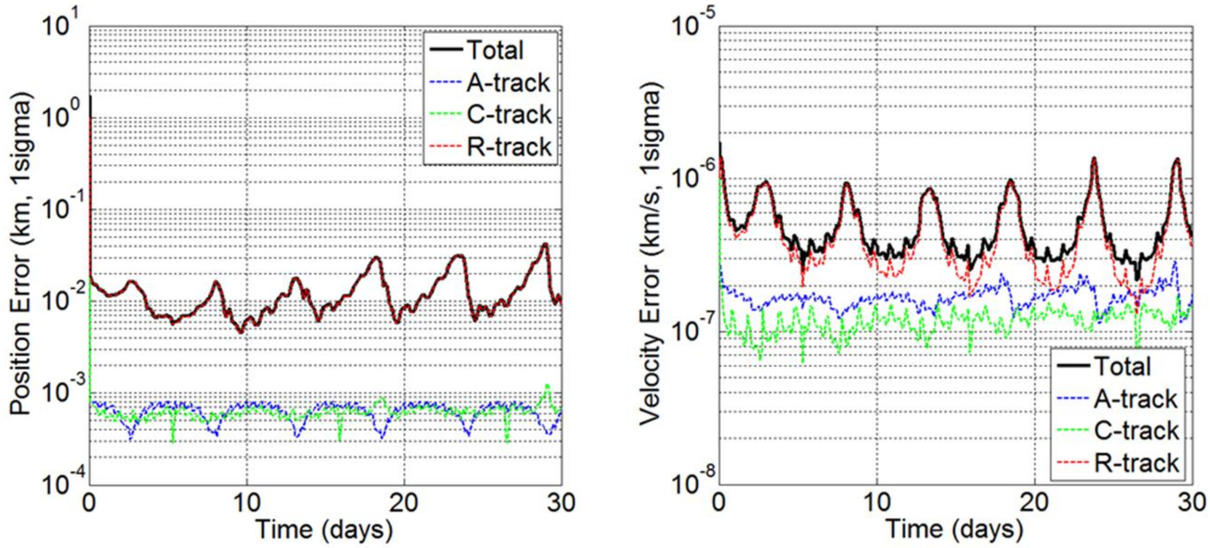

Figure 3. Asteroid-S/C state estimation with radiometric tracking, LoS and altimetry

Figure 3 shows the estimation performances achieved by including also altimetry measurements in the simulation. Altimetry provides direct information on the radial position component, thus improving significantly the estimation also of the along-track velocity (directly depending on the distance and body's gravitational parameter). The error levels achieved with radiometric data, optical measurements and altimetry are of 10-20 m radial position, 1 m along-track and cross track position, and .1-1 mm/s radial velocity and .1-.2 along-track and cross-track velocity, all 1-sigma.

The asteroid ephemerides are being estimated in the process. More information regarding ephemerides estimation is provided in Section 3.2.

3.1. Asteroid gravity field estimation

A fundamental part of any small body exploration mission is the estimation of the body's gravity field. Such estimation is accomplished by including the parameters describing the gravity in the estimation process. For this purpose sufficiently long arcs with stable uncontrolled motion are required to avoid polluting the radiometric tracking data with manoeuvres' noises.

In the simulations performed for the previous section of this paper no manoeuvres were included, assuming that the S/C is flying an orbit stable for the duration of the simulation. Long uncontrolled stability times can be achieved on orbit configurations like Sun-Stabilized Terminator Orbits (see [6]), but it is possible that correction manoeuvres will be required in actual operational scenarios to compensate for trajectory errors. In such conditions the radio-tracking experiment for gravity determination shall be carried out over several arcs, separated by manoeuvres. The simulation of such detailed operational conditions goes beyond the scope of the work carried out for this paper; in addition the simulation of a continuous tracking arc is sufficient to assess what are the realistic achievable knowledge levels for the gravity parameter of the asteroid in real mission conditions.

Figure 4 shows the performances that can be achieved in the estimation of the gravitational parameter of 2008 EV5 on uncontrolled orbits with 2.5 and 5 km of altitude from Phobos. The inclusion of altimetry measurement does not appear to change substantially the performances, and, as expected, reducing the distance from the gravity source a much more accurate knowledge can be reached. The final estimation errors are in the order of 3E-10 km^3/s^2 for orbits at 5 km and 2E-11 km^3/s^2 (both 1-sigma), corresponding to respectively 5% and 0.3% of the nominal value of the gravitational parameter.

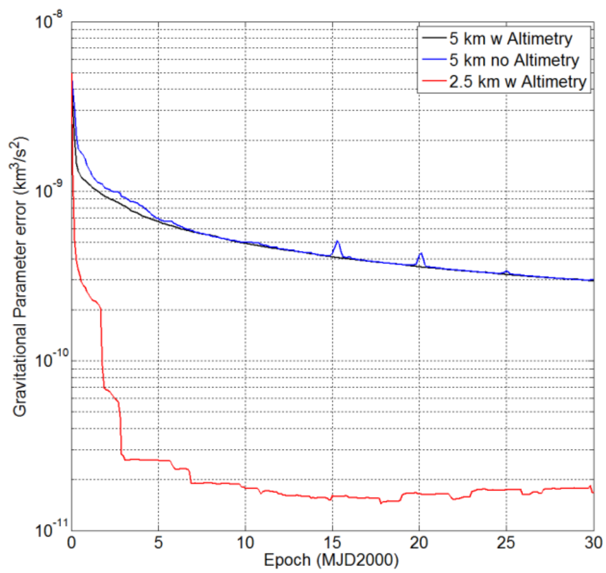

Figure 4. Gravitational parameter estimation performances

3.2. Asteroid ephemerides estimation

The accuracy in the estimation of the asteroid ephemerides will be determined by the accuracy in the estimation of the Earth-S/C state and of the asteroid-S/C state. As seen in previous sections, a good knowledge of the asteroid-relative state (position error in the order of 1-10 m 1-sigma) is achieved making use of range, Doppler, camera and altimetry data. In order to significantly improve the ephemeris estimation also Delta-DOR measurements need to be included in the estimation process. This is due to the fact that Delta-DOR data provide precise information on the S/C position in plane of sky, which allow a relevant improvement in the accuracy of the Earth-S/C state knowledge, which will be the main contributor to the knowledge error for the case in which accurate optical and altimetry measurements are available.

Notice that DDOR data, due to the small motion of the S/C in the plane of the sky on its orbit around the asteroid, do not appear to provide major improvements to the estimation of the asteroid-S/C state. In real operations the ephemerides of 2008 EV5 will be improved progressively as the S/C approaches the asteroid thanks to the optical LoS data, up to the acquisition of an asteroid-bounded orbit, on which the knowledge of 2008 EV5 ephemerides is expected to reach the maximum accuracy. In this section the same 5 km altitude orbit employed previously will be considered.

Table 3 summarises the Delta-DDOR measurements assumptions employed in the process. Figure 5 shows the achieved ephemerides knowledge level (1-sigma). Initial errors were set to 1000 km in position; the filter quickly converges from its initial arbitrary errors to the regime estimation accuracy allowed by the measurements. With no DDOR data the asteroid position knowledge is approximately 100 km (1-sigma), while including DDOR a substantial improvement down to levels of 1 km (1-sigma) is achieved.

Table 3. Delta-DOR measurements assumptions

DDOR Measurements		
	Noise (1-sigma)	Bias (1-sigma)
Measurement noise	0.05 m	-
Ground station position (error per component)	-	5 cm
Baselines Considered	Cebreros-New Norcia, Cebreros-Malargüe	
Acquisition frequency	Daily from each baseline	

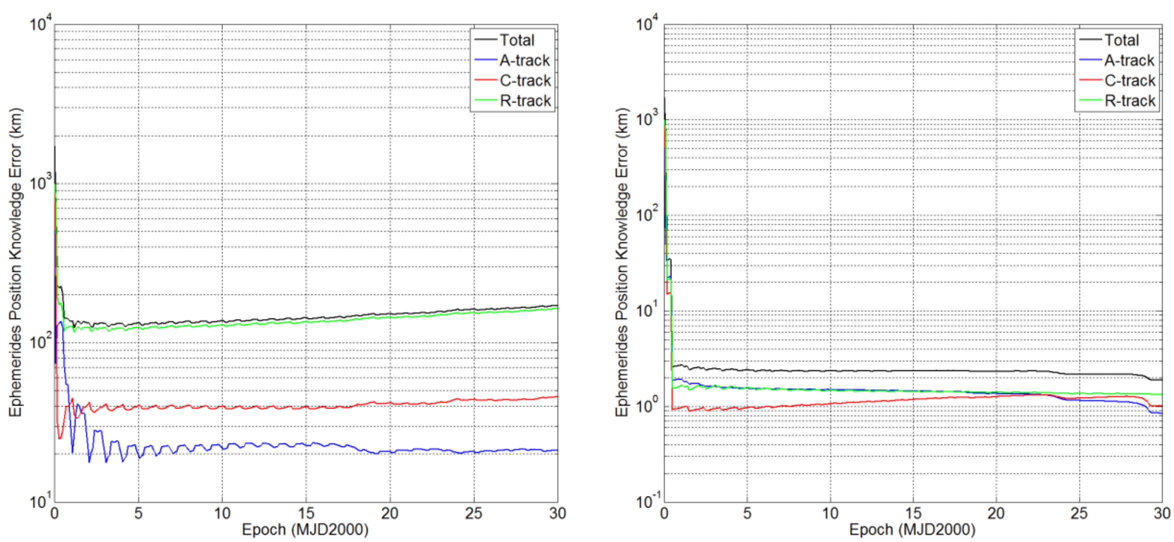

Figure 5. 2008 EV5 ephemerides knowledge (1-sigma) without (left) and with (right) DDOR measurements

3.3. Impact of operational delay

Trajectory guidance for the proximity phase was based on a ground-in-the-loop scheme, meaning that measurements are processed on ground to estimate the S/C state and compute the required trajectory guidance manoeuvres.

Due to the time required to process the measurements, estimate the state, and compute the manoeuvres, a delay due to operations must be taken into account between the last useful measurement and a burn execution. This operational delay will define the time at which the latest useful measurement can be acquired before a manoeuvre; the tracking data collected after the measurement cut-off time will then be available again for the guidance after the upcoming burn.

The accuracy with which the state is known for guidance computation will be obtained from the accuracy at the time of the last useful measurement propagated forward without including measurements. Figure 6 shows the accuracy in knowledge of the state at time of manoeuvre execution as function of the operational delay for a circular polar orbit at 5 km altitude. The knowledge accuracy for the state at the time of the last acquired measurement is assumed to be as obtained from the simulations of Section 3 with altimetry, thus: 1 m in along-track and cross-track position, 10 m in radial position, 0.1 mm/s in along-track and cross-track velocity, and 1 mm/s in radial velocity.

The plots show how the state estimation accuracy is worsened up to 300 m and 7 mm/s (1-sigma) after 24h of operational delay time.

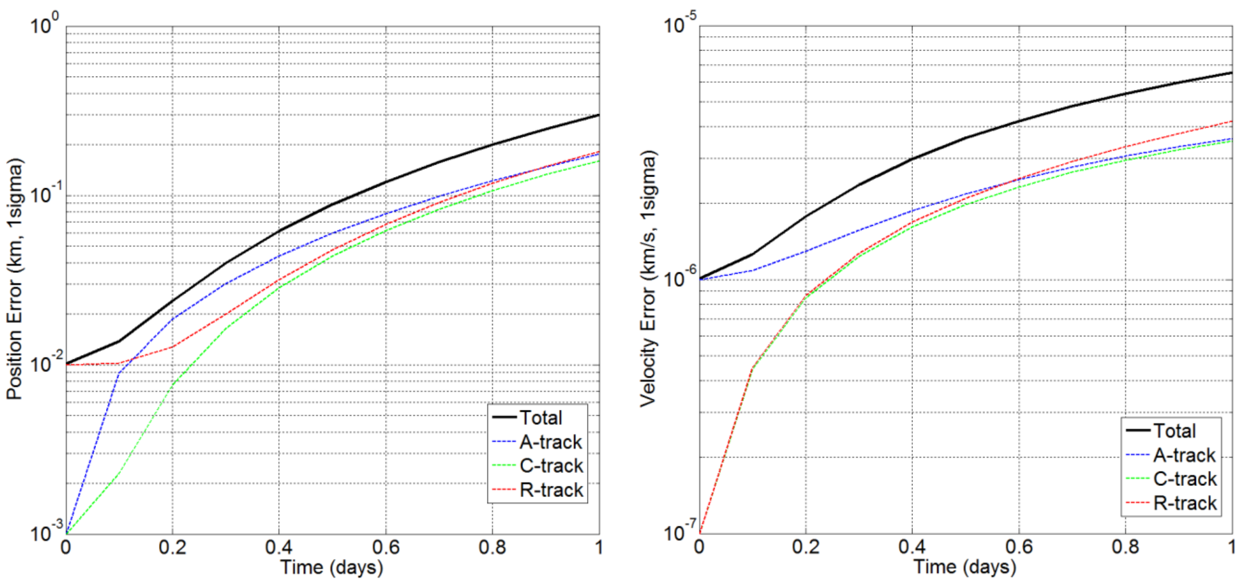

Figure 6. Evolution of asteroid-S/C state knowledge during operational delay

4. Guidance at asteroid proximity

Simulations were executed to assess the trajectory guidance performances that can be achieved at 2008 EV5 proximity for the 5 km polar orbit previously mentioned. Monte Carlo shots were executed over a time span of 5 days, assuming that a ground-based guidance strategy is employed. The measurement set is as already discussed in the navigation analysis section, including ground-processed altimetry and optical LoS measurements. Daily control manoeuvres were scheduled, targeted to the nominal position at the end of the 5 days-long arc. An additional manoeuvre was added 5 hours before the targeting time to represent a fine steering to the de-orbiting time, assumed to be located at the end of the simulated arc. This final manoeuvre could be executed for example to target the S/C to the de-orbiting start point. The trajectory burns were simulated with the assumptions described in Table 4.

Table 4. Assumptions for guidance manoeuvres

Guidance Assumptions	
TCM Execution and estimation errors	2% size, 1.5 deg direction (3-sigma)
Operational delay	12 h

Figure 7 shows the results obtained: dispersion errors are in the order of 100 m in position and 10 mm/s, while the manoeuvres have sizes in the order of 10-20 mm/s (99 %ile).

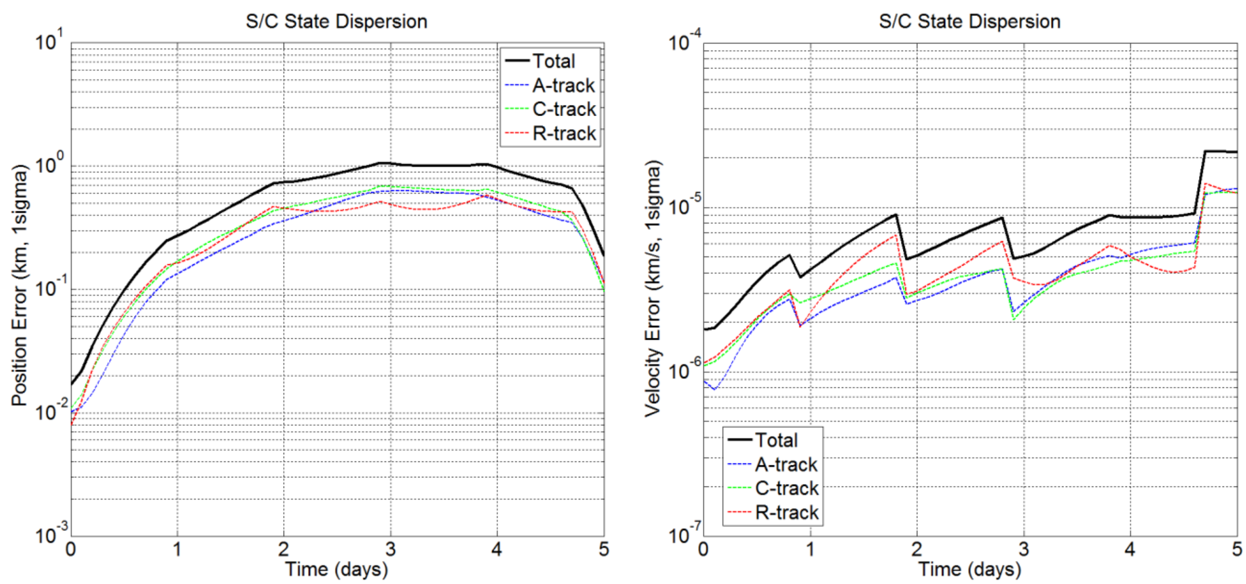

Figure 7. Guidance performance for 5 km altitude circular polar orbit

4.1. Radial descent

In this section guidance results for radial descent down to 100 m of altitude above the asteroid surface are reported. The descent is assumed to be started by zeroing out the inertial velocity of the S/C with respect to the asteroid; as a consequence, the S/C will start falling towards the asteroid. At the end of the radial descent, a vertical descent sequence is started, in which the S/C is actively controlled to achieve a null lateral velocity at touchdown with respect to the target surface landing site. This last vertical descent phase is not analysed in this paper.

Figure 8 shows the free fall time at 2008 EV5 as function of the initial altitude, and for final altitude 100 m above a spherical surface. As it is possible to see from the figure, due to the low gravity of the asteroid, for an initial altitude of 4 km more than a day is required to complete the descent. Such long descent duration will allow executing ground-based guidance during the descent to reduce the dispersions at the end of the descent. For short descent time either no guidance can be performed, or an autonomous GNC architecture would be required. This last option was discarded during the study to minimise the cost and complexity of the mission.

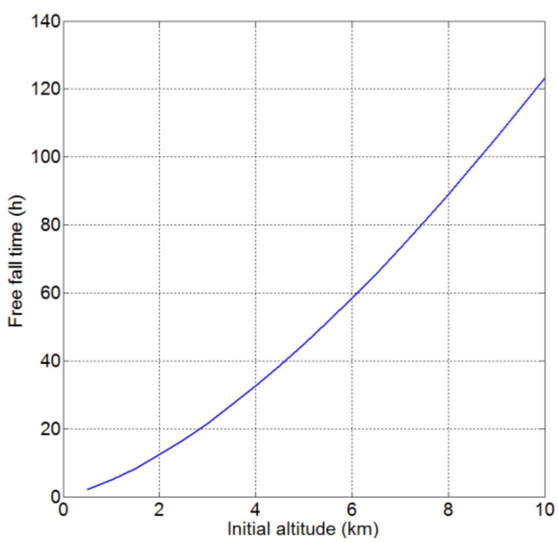

Figure 8. Free fall time at 2008 EV5

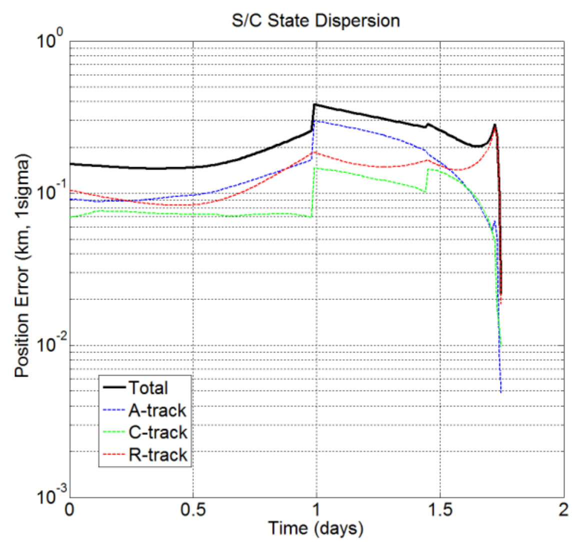

Figure 9. Position dispersion in radial descent

For the descent branch radiometric and optical+altimetry measurements were considered. At sufficiently large distances optical Line of Sight data to the body centre of brightness could be employed. Getting closer to the asteroid, its size in the camera FOV will become larger, and surface characteristics will become clearly visible. A detailed navigation analysis requires complex modeling of the many factors involved in the process. In line with the early study phase in which they were executed, the optical navigation discussion presented here is based on the use of performance models for the measurement types considered. Landmark measurements allow estimating the position of the S/C in an absolute asteroid-fixed frame. The landmark extraction and associated S/C position estimation is a very complex task, which is out of the scope of the mission study presented here. As a consequence, landmark tracking measurements performances were simulated assuming that they provide absolute information on the line of sight from the S/C to 2008 EV5, with associated errors depending on the camera characteristics and on the distance to the surface. The logic behind the simulation of this type of measurements, and the sizing of the associated working range and measurement errors were based on the work presented in [7]. Optical measurements are assumed to be taken in this phase with a Narrow Angle Camera with 1024 px and 2 deg of Field of View. Images are acquired every hour and sent to ground for processing. Measurement noise was set to 3 px (1-sigma). The knowledge error for the asteroid rotation, relating the surface-relative position and the inertial position, are assumed to be negligible, as descent will be executed after a long asteroid characterization phase.

The touchdown errors will be driven by the residual velocity errors after the execution of the last TCM, which will depend on the size of the TCM itself. The size of the TCM will depend on the initial errors at beginning of descent.

Assuming that the S/C is initially orbiting the asteroid at 5 km distance from the surface, the expected dispersion and knowledge errors before de-orbiting can be derived from the information provided in the above sections. The dispersions included by the de-orbiting itself will again be driven by the size of the de-orbiting. For the selected 5 km initial altitude, on a circular orbit the S/C would have an orbital velocity of approximately 35-40 mm/s. As the initial de-orbiting burn is meant to cancel the inertial velocity of the S/C with respect to the asteroid, that would be the size of the de-orbiting manoeuvre. With manoeuvre errors as the ones considered in Table 4, the residual velocity errors at beginning of descent would be in the order of few mm/s. As a consequence, the larger contribution to dispersion errors over the de-orbiting and radial descent would be the velocity dispersions before de-orbiting execution, which are in the order of 10 mm/s (1-sigma) in each direction.

A quite demanding descent timeline was needed to achieve acceptable performances at the end of the radial descent. Three TCMs are scheduled: one day after de-orbiting, 7.5 hours before end of the de-orbiting, and 0.5 hours before end of de-orbiting. Dispersions at the end of the simulation are in the order of 20 m radial, and 12 m in plane (alongtrack-crosstrack, 1-sigma), as shown in Figure 9.

4.2. Inertial hovering

Inertial hovering is an asteroid proximity strategy which ideally maintains the S/C at a given altitude along a given inertial (or pseudo-inertial) direction. The practical implementation of the guidance is based on the definition of a control box, whose boundaries can be a minimum asteroid altitude, or a maximum angle with respect to the given inertial direction, or by a combination of the two.

Inertial hovering is suitable for asteroid observations as it can allow continuously looking at its illuminated side. Controlled polar orbits will spend half of their period above the side of 2008 EV5 opposite to the Sun; due to the long orbital period caused by the reduced gravity, this makes controlled polar orbits to be inefficient for surface observation.

In the specific case analysed here it is assumed that the hovering will be keep the S/C along the asteroid-Sun line; such direction is not inertial due to the motion of the asteroid in the solar system. The motion of such reference direction is slow, but it needs to be taken into account in the simulations as it will affect the trajectory. The actual guidance logic is thus based on the S/C moving along an elliptic arc centred along the reference direction: when one of the active boundaries is hit, a manoeuvre is executed such to invert the velocity (thus, rebounding the S/C towards the interior of the control box), and to target the S/C towards the reference direction by forcing the trajectory plane to include the reference direction. Notice that for a given semi-major axis and eccentricity the angle and altitude control boxes are not independent.

For a given hovering altitude, the guidance cost strongly depends on the nominal eccentricity of the orbital arc along which the S/C is moving. Increasing the eccentricity with constant semi-major axis a lower delta-V is needed and the time between controls is increased, but the pericentre altitude becomes lower, meaning that a missed control will require safe orbit acquisition to avoid impacting the asteroid.

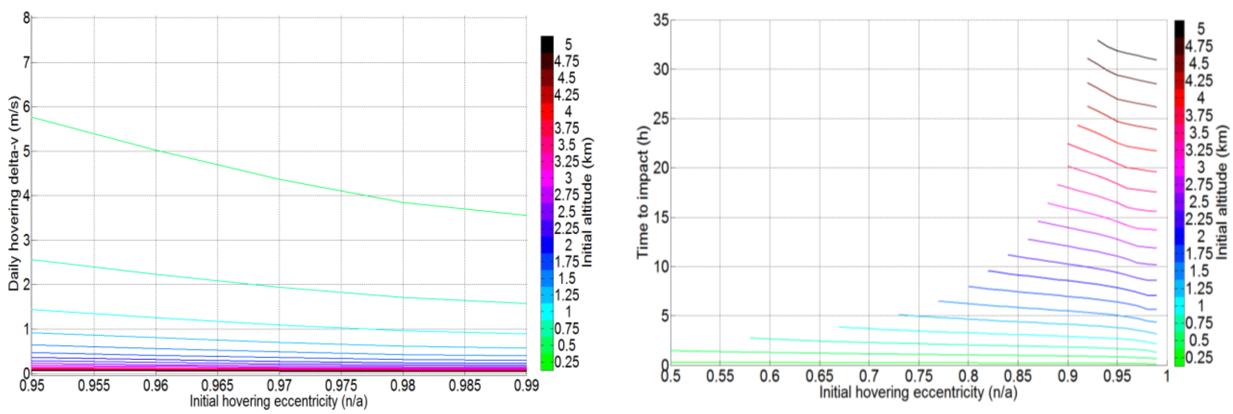

Figure 10. Daily Hovering deltaV and Time to impact vs initial eccentricity

The plots in Figure 10 show the daily hovering delta-V and time to impact in case of missed control, for perfect guidance and navigation, as function of the initial eccentricity for different inertial hovering altitudes above asteroid 2008 EV5. For eccentricity above 0.9 and initial altitude above 2.5 km, the time to impact is above 10 hours, leaving sufficient time for collision avoidance manoeuvre execution in case of missed burn. The time between manoeuvres is above 5 hours for initial eccentricity higher than 0.95 and initial radius above 2.5 km.

The guidance for the simulations carried out is assumed to be autonomous, thus with no operational delay. Optical LoS and altimetry measurements are acquired and processed on-board to compute the necessary manoeuvres; no radiometric data were considered. Navigation was included in the loop in the simulations, with measurements simulated making use of performance models fed to a square root information filter to obtain an estimation of the state. Measurement assumptions are as in Table 2. Control is represented by impulsive velocity changes. Errors in the execution of the burns are taken as in Table 4.

In addition to the altitude/angle control box, it is possible to set the minimum time between subsequent controls. A minimum control-free duration could be enforced to allow acquisition of asteroid images without the noise on the S/C induced by the thrusters; in addition, if a successful guidance can be achieved with a sufficiently long time between subsequent burns, ground-based operations could be employed for the inertial hovering phase, provided that the estimation performances including the operational delay are sufficient.

Monte Carlo simulations were executed for 7 days of hovering, with 200 m maximum altitude variation, and 5 deg of maximum angular excursion with respect to the hovering direction (here taken as the asteroid-Sun line).

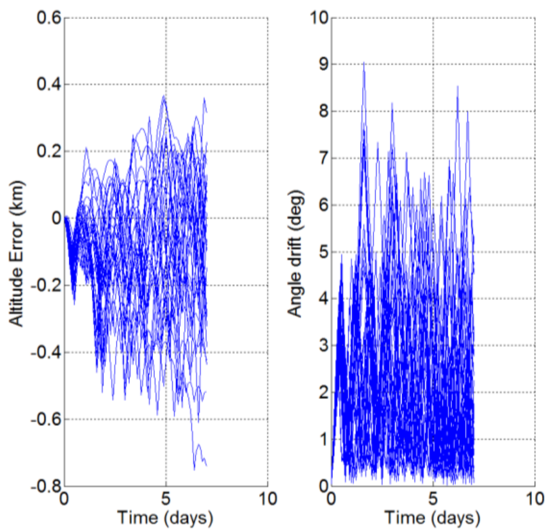

Figure 11. Inertial hovering with 8 hours between burns.

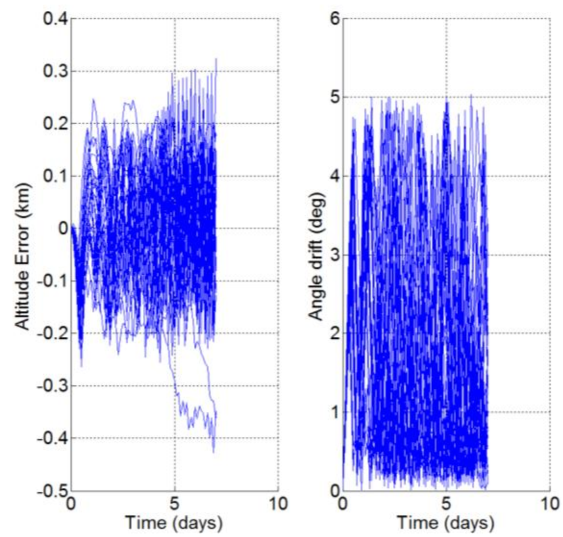

Figure 12: Inertial hovering with 1 hour between burns.

Figure 11 and Figure 12 show the results obtained by assuming a minimum time between controls of 8 hours and 1 hour; in the case of 8 hours the maximum altitude variation recorded is below 800 m, while the maximum angular displacement is 9 deg. For 1 hour between burns the guidance provides much better performances, almost respecting the assigned boundaries, with maximum altitude variation of 300 m and maximum angular excursion of 5 deg.

It is anyway important to point out that the above results were obtained with a fine tuning of the involved parameters. The design and testing of the inertial hovering GNC was purely preliminary, and would require further work to achieve a more robust behaviour.

5. Conclusions

The results obtained in the preliminary study for the MarcoPolo-R mission, presented in this paper, allowed providing substantial support to the design of the whole mission architecture.

The asteroid-S/C state can be estimated to errors in the order of 1-10 m (1-sigma) making use of radiometric, optical and altimetry data. The asteroid ephemerides knowledge will have the same accuracy as the Earth-S/C state knowledge; including Delta-DOR measurements in the estimation process the ephemerides can be estimated down to errors of 1 km (1-sigma) in each component.

The current uncertainty in the asteroid mass is in the order of 50%; executing radiometric tracking for a 5 km altitude orbit the gravitational parameter accuracy is improved to errors of 5%, and reducing the altitude to 2.5 km the estimation is improved to .5% (all 1-sigma).

Guidance performances are such that errors in the order of 100 m in each component (1-sigma) can be achieved for the asteroid-S/C position; radial descent was simulated with ground-based guidance, with 3 TCMs, and allowed achieving 12 m of error (1-sigma) in the plane perpendicular to the radial direction.

Inertial hovering was simulated with the assumption of autonomous GNC, and errors of 200 m in altitude and 5 deg with respect to the reference direction were found after 5 days of hovering (3-sigma).

Further work would be required to refine all the presented data, and especially in the design and analysis of the inertial hovering phase, in order to increase the performances and robustness GNC logic implemented.

6. Acknowledgments

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7. References

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